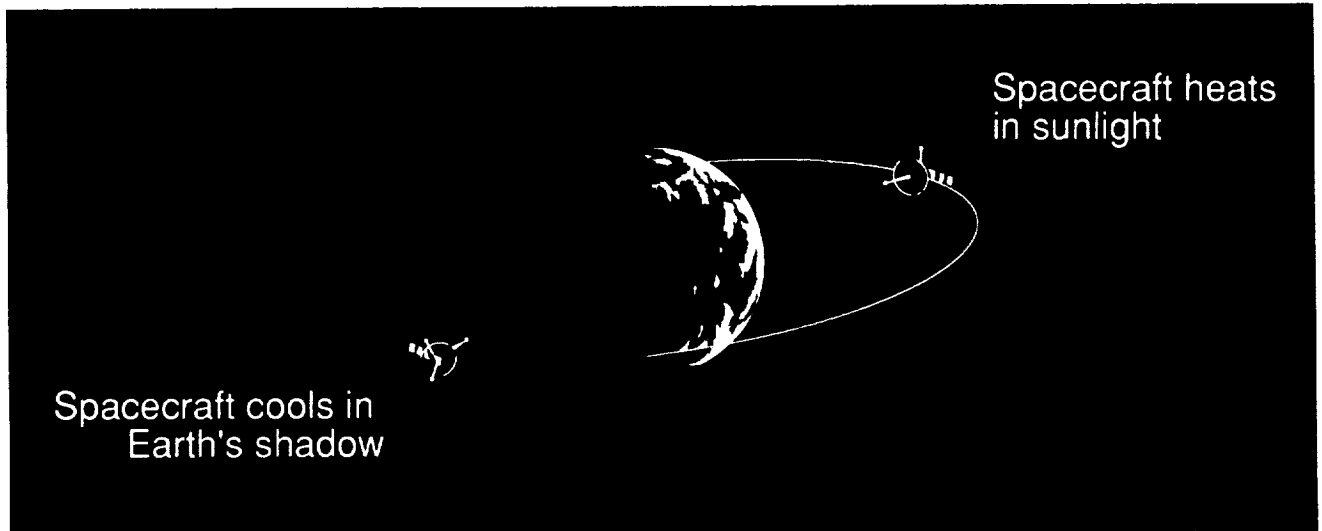


**EFFECTS OF THERMAL CYCLING ON COMPOSITE MATERIALS FOR
SPACE STRUCTURES**

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THERMAL CYCLING OF COMPOSITE MATERIALS IN ORBIT

Thermal cycling, which can result from the spacecraft orbiting the Earth, is one of several space service environmental parameters that can affect composite structural materials, ref. 1. As the spacecraft passes in and out of the Earth's shadow, the temperature of the structure rises and falls. The minimum and maximum temperatures reached and the induced effects on the material are directly related to the properties of the material and the thermal control coating. The materials may also experience thermal cycling as a result of structural members casting shadows on other parts of the structure.



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OBJECTIVE

The objective of this paper is to briefly describe the effects of thermal cycling on the thermal and mechanical properties of composite materials that are candidates for space structures. The outline for this paper is shown below. The results from a thermal analysis of the orbiting Space Station Freedom will be used to define a typical thermal environment and the parameters that cause changes in the thermal history. The interactions of this environment with composite materials will be shown and described. The effects of this interaction on the integrity as well as the properties of Gr/thermoset, Gr/thermoplastic, Gr/metal and Gr/glass composite materials are discussed. Emphasis will be placed on the effects of the interaction that are critical to precision spacecraft. Finally, ground test methodology will be briefly discussed.

EFFECTS OF THERMAL CYCLING ON COMPOSITE MATERIALS FOR SPACE STRUCTURES

OUTLINE

- Thermal environment
- Material/environment interaction
- Effects on materials:
 - Gr/thermoset
 - Gr/thermoplastic
 - Metal - matrices
 - Glass - matrices
- Ground test methodology

SPACE STATION FREEDOM TRUSS STRUCTURE THERMAL ANALYSIS

The expected thermal history of the Space Station Freedom truss structure will be used to illustrate a typical thermal input to a spacecraft. The temperature history of a single 2-inch diameter, P75 graphite/epoxy tube orbiting the earth at 270 nautical miles was analyzed with the TRASYS orbital mechanics model and the SINDA thermal analyzer assuming no shadowing by other structural members. The tube temperature history is shown below.

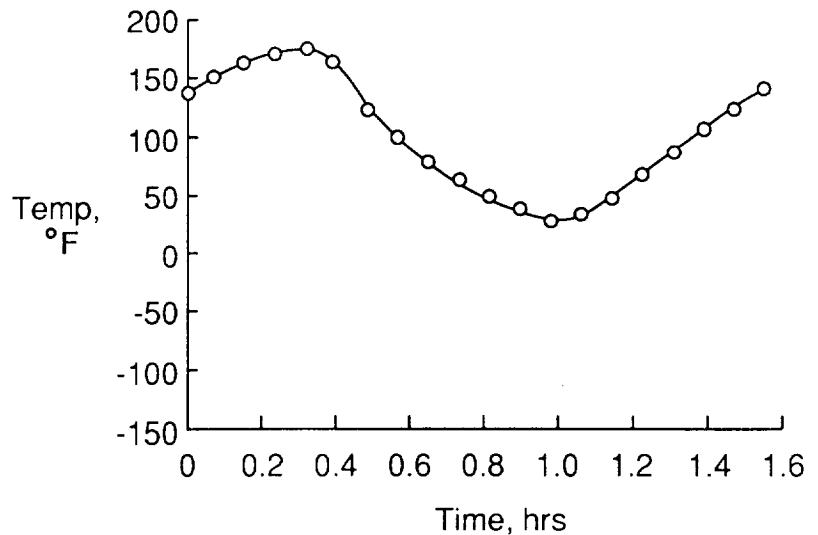
For a tube with a solar absorptance-to-emittance ratio, α_s/ϵ , of 0.3/0.2, the temperature varies between about 175°F to about 25°F over a period of about 90 minutes. The period of the thermal cycle depends upon the altitude of the orbit and the amplitude of the cycle depends upon the α_s/ϵ ratio.

Thermal analysis

- Single tube (9 ft long, 2 in. ID)
- P75 graphite/epoxy
- Orbital parameters
 - Beta = 0 deg.
 - Altitude = 270 N. miles
- TRASYS and SINDA models

Typical tube temperature cycling range

Solar absorptance/emittance = .3/.2

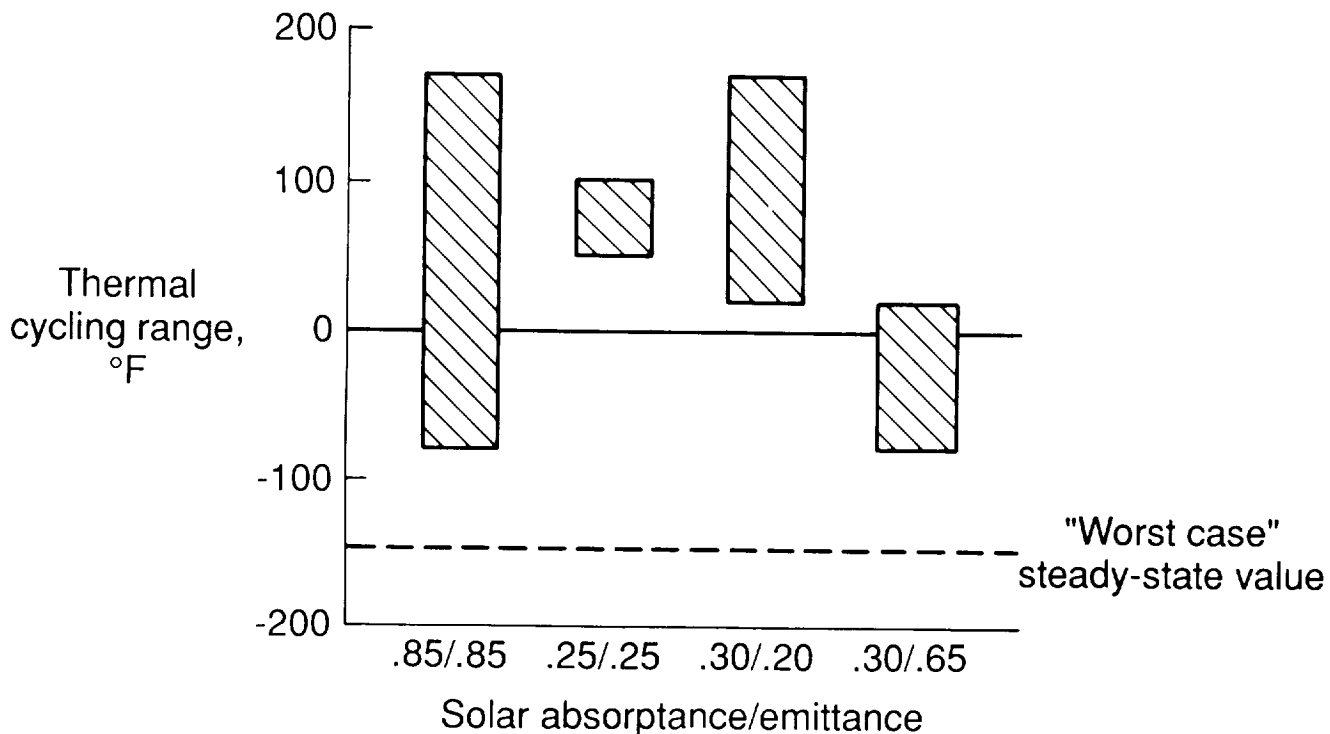


EFFECTS OF SOLAR ABSORPTANCE/EMITTANCE ON TEMPERATURE RANGE

The effects of material surface solar absorptance/emittance ratio, α_s/ϵ on the amplitude of the thermal cycle is shown in the figure below. These results are from a transient analysis of a 2-inch diameter, P75S Gr/Ep composite tube orbiting the Earth. The data shows that the temperature range is very sensitive to α_s/ϵ . For a bare Gr/Ep tube, $\alpha_s/\epsilon = 0.85/0.85$, the temperature range is from about 175°F to about -80°F. The ideal range would be small and centered around room temperature. However, degradation of the thermal control coating could change the α_s/ϵ ratio and significantly change the range. For example, degradation from 0.25/0.25 to 0.30/0.20, causes the temperature range to change from 50°F to 100°F to about 20°F to 175°F. For Space Station Freedom the steady-state cold temperature is estimated to be -150°F; however, some spacecraft could reach as low as -250°F.

SPACE STATION TRUSS STRUCTURE THERMAL CYCLING RANGE

Single 2 in. i.d., P75 Gr/Ep tube



COMPOSITE CONSTITUENT AND LAMINA PROPERTIES

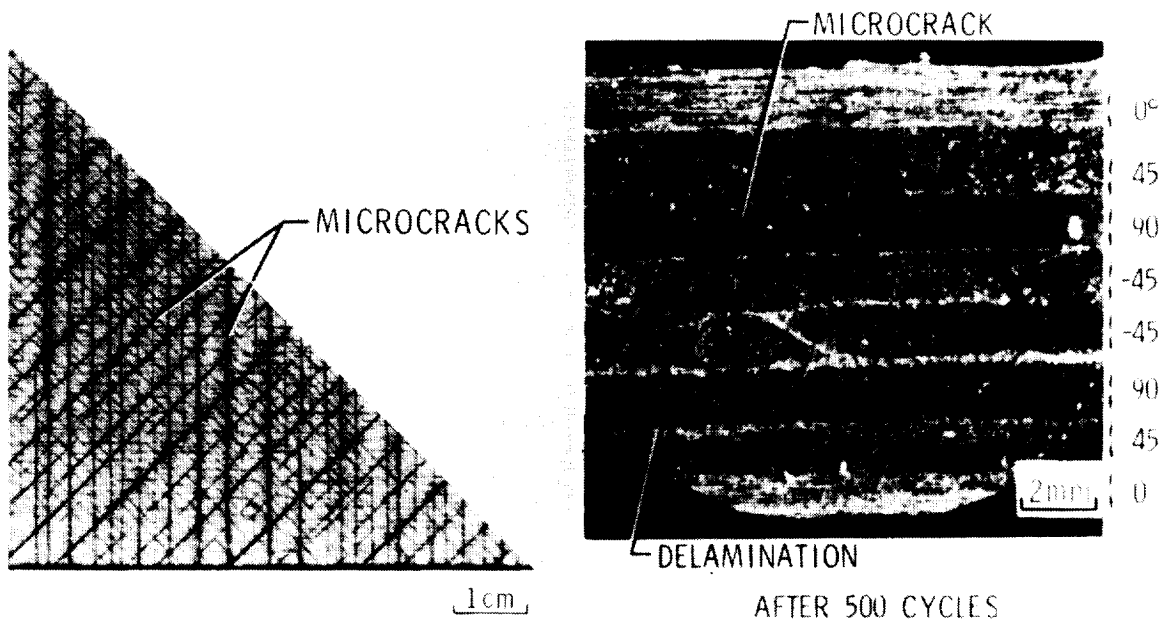
Composite materials are made by combining fibers and matrix materials to form a simple lamina. Laminae are then stacked in various ways to form laminates with the desired properties. Typical room temperature thermal and mechanical properties of continuous graphite fibers, matrix and laminate considered for space structures are shown below. Both high strength and modulus fibers are candidates. The orthotropic character of the fibers becomes evident when the axial and radial properties are compared. Note that the axial coefficients of thermal expansion (CTE), A_1 , are small and negative. The matrix materials have low strength and low modulus and, except for the glass, have large, positive CTE. When combined, the fibers and matrices form highly orthotropic unidirectional lamina. The large differences in the CTE of the fiber and matrix as well as the differences in the directional CTEs of the lamina can result in very high stresses induced by thermal cycling in the laminates as well as the lamina.

Material	Modulus		Strength		CTE	
	E1 Msi	E2 Msi	Xt ksi	Yt ksi	A1 ppm/F	A2 ppm/F
Fibers						
T300	33.8	3.35	350	--	-0.30	5.6
HMS	55.0	0.90	250	--	-0.55	3.80
P75	79.8	1.38	300	--	-0.75	3.80
P100	115.5	1.05	325	--	-0.78	3.80
Matrix						
934	0.67	--	8.5	--	24.4	--
PMR 15	0.5	--	8	--	20.0	--
2024 Al	10.6	--	60	--	12.9	--
Glass	9.1	--	--	--	1.8	--
Lamina						
T300/934	18.9	1.4	223	9.37	-0.01	16.13
P75/934	42.0	0.83	102	3.51	-0.58	19.18
HMS/Glass	24.	1.1	86.3	2.6	-0.23	2.1
P100/2024	47.8	3.6	92.2	8.89	0.800	14.51

TYPICAL THERMAL INDUCED MICRODAMAGE

The primary effect of thermal cycling on composites is the induced thermal stresses and the resulting damage. Typical microdamage induced by thermal cycling of continuous graphite reinforced polymers is shown in the figure below. The micrograph shows microcracks and delaminations formed in a $[0/45/90/-45]_s$ laminate of C6000/PMR-15 after 500 cycles between -156°C and 316°C , as viewed along a polished edge of the specimen. The thermal cycling in orbit can be considered low amplitude thermal fatigue with the net material effects (microdamage) changing with time. The microcracks began as intraply cracks and grew to interply cracks as the number of cycles increased. Delaminations appeared as the number of cycles increased. An X-ray of the top of the specimen is also shown below. The dark lines are the cracks, enhanced with a dye, that run parallel to the fibers in each lamina over the entire width and length of the specimen. Although this composite material is very brittle, has a high cure temperature (resulting in high residual stresses), and the temperature range is large, similar cracks have been observed in composites with lower cure temperatures even after cycling between a smaller temperature range, -100°C to 66°C .

TYPICAL X-RAY OF C6000/PMR-15 LAMINATE [0/45/90/-45] AFTER 500 CYCLES BETWEEN -156°C AND 316°C



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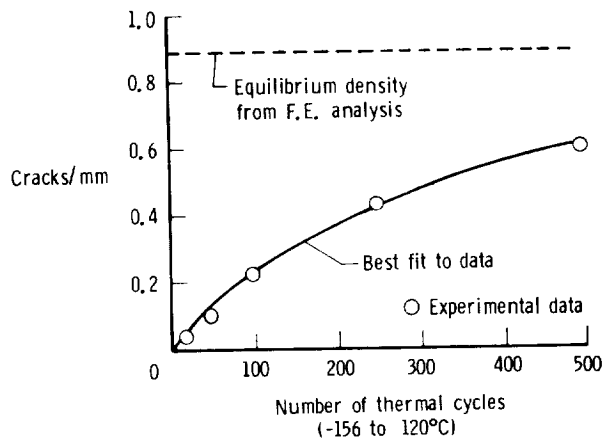
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THERMAL CYCLING EFFECTS ON GRAPHITE/EPOXY COMPOSITES

The increase in the crack density (cracks/length of specimen) as thermal fatigue progresses, is shown below. As the number of cycles increases, the crack density asymptotically approaches an equilibrium value. For the T300/5208 $[O_2/90_2]_s$ graphite/epoxy laminate shown here, the equilibrium density may not be reached for several thousand cycles. The coefficient of thermal expansion, CTE, is one of the laminate properties that can be greatly affected by microcracking. For this laminate, the CTE was reduced by about 40% after 500 thermal cycles. Additional reduction may be expected since the laminate had not reached its equilibrium crack density for this temperature range.

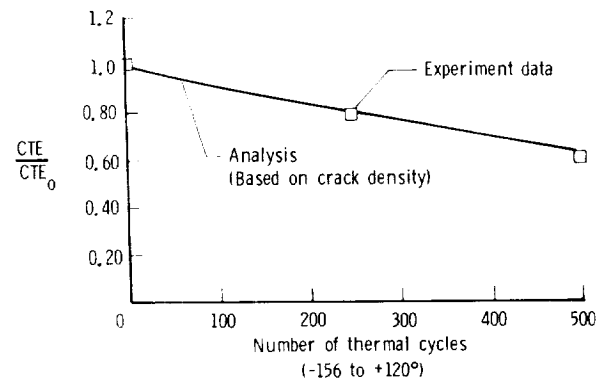
EFFECT OF THERMAL CYCLING ON MICROCRACK DENSITY

$[O_2/90_2]_s$ T300/5208 Gr/Ep



EFFECT OF THERMAL CYCLING ON THE COEFFICIENT OF THERMAL EXPANSION

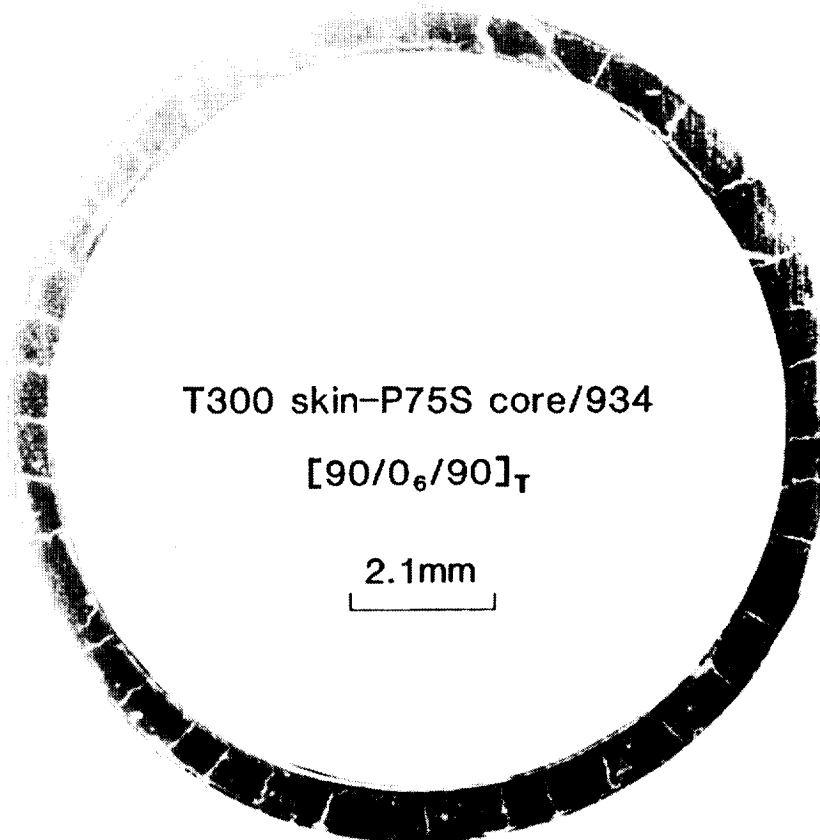
$[O_2/90_2]_s$ T300/5208 Gr/Ep



MICRODAMAGE IN GRAPHITE/EPOXY TUBES

The data discussed in the previous figures has been for flat specimens. Microdamage can also be induced in composite tubes. The cross section of a 0.5-inch diameter composite tube that was cycled 500 times between -156°C and 94°C is shown below. The radial cracks induced in the wall by the thermal cycling can be clearly seen. Some delamination along the inside and outside diameters was also induced. After 500 cycles, the crack morphology is very similar to that of the flat laminates.

MICRODAMAGE IN Gr/Ep TUBE AFTER 500 THERMAL CYCLES BETWEEN -156°C AND 94°C

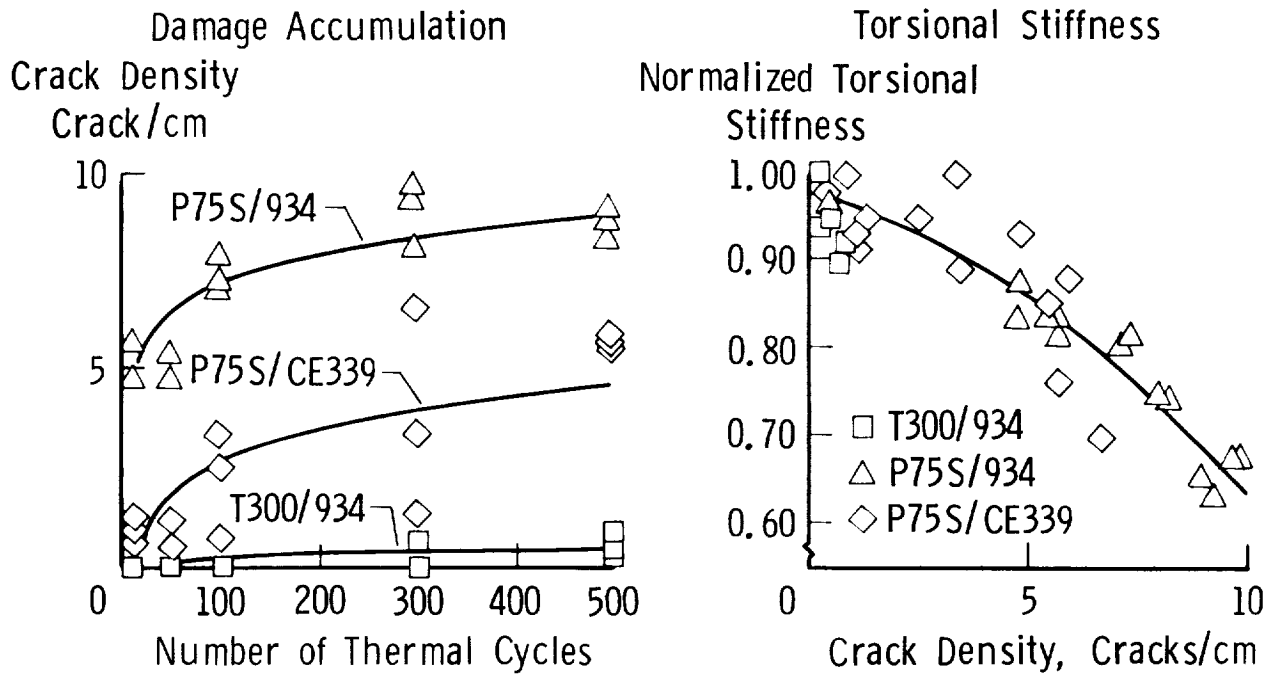


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EFFECTS OF THERMAL CYCLING ON COMPOSITE TUBES

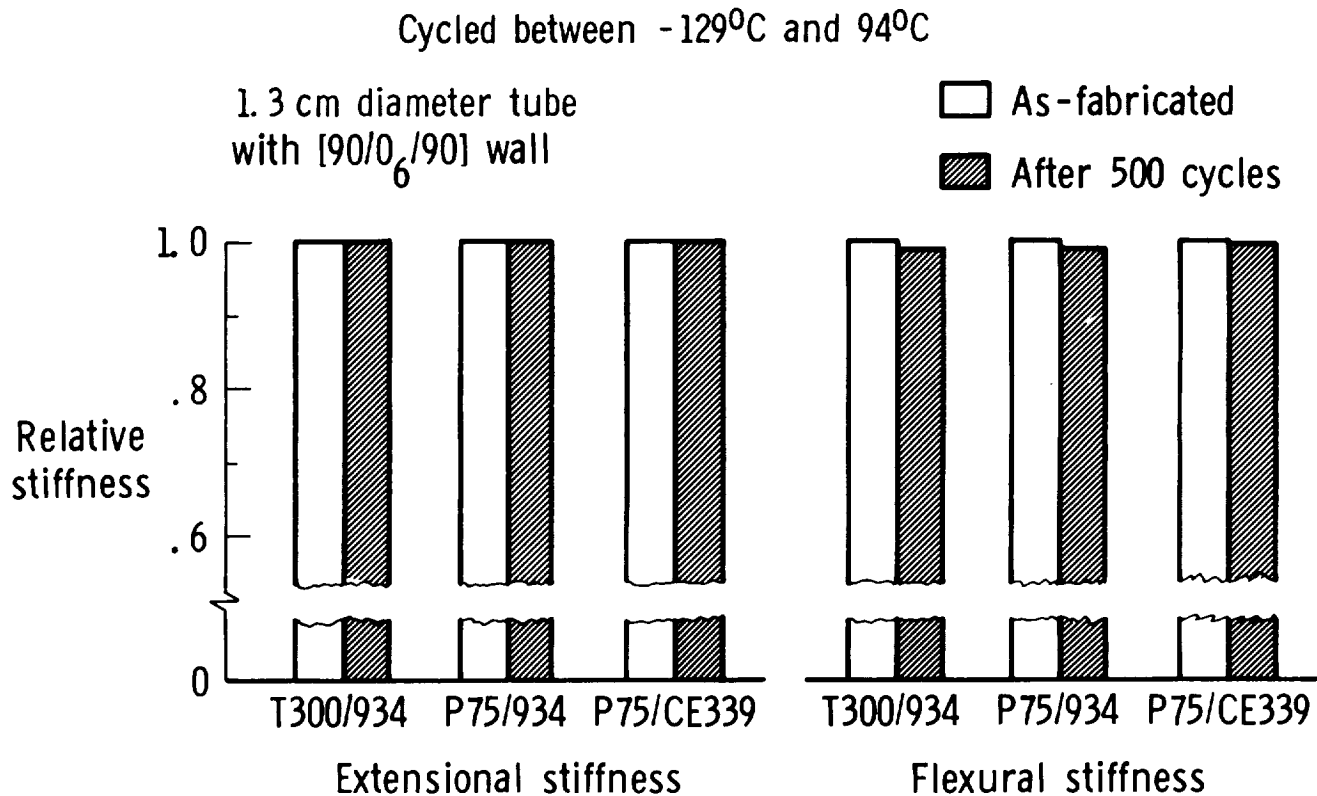
The change in crack density induced in three tubes of different materials, as the number of thermal cycles between -156°C and 94°C increases, is shown in the figure below. The P75S/934 is a high modulus brittle epoxy system, the P75S/CE339 is a high modulus toughened epoxy system, and the T300/934 is a low modulus brittle epoxy system. The crack densities for each material asymptotically approach equilibrium values as the number of cycles increases. The effects of the thermal cycling or microcracking on the torsional stiffness of these tubes are also shown below. The torsional stiffness of tubes of each of the three materials was reduced by about 40% and the change in the stiffness appeared independent of the composite material system. These data illustrate the sensitivity of matrix dominated properties to microcracking.

(Cycled Between -156°C and 94°C)



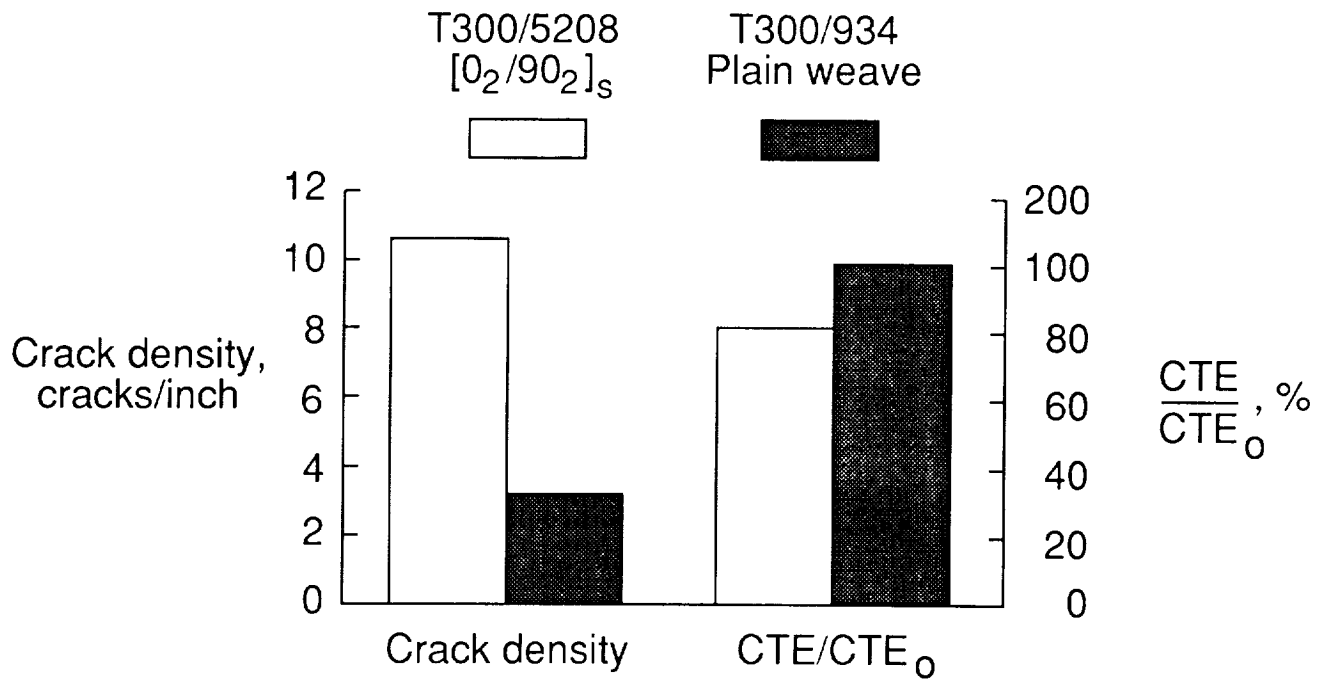
EFFECTS OF THERMAL CYCLING ON EXTENSIONAL AND FLEXURE STIFFNESSES OF Gr/Ep TUBES

The extensional and flexural stiffnesses of the three different Gr/Ep tubes before and after 500 thermal cyclings between -129°C and 94°C are shown below. These are the same materials shown in the previous figure. The data show no significant effects of thermal cycling on these stiffnesses; this was expected since, for this wall configuration, the extensional and flexural stiffnesses are fiber dominated properties, which are not sensitive to matrix damage.



EFFECTS OF FABRIC ON THERMAL CYCLING STABILITY OF Gr/Ep COMPOSITES

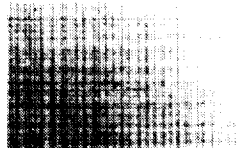


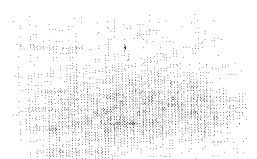
The results from one study to minimize microdamage induced by thermal cycling., ref. 2, are shown in the figure below. Crack densities and changes in CTE for a continuous graphite $[0_2/90_2]_s$ laminate of T300/5208 and a laminate of plain weave T300/934 Gr/Ep are compared. The 5208 epoxy resin and the 934 epoxy resin are very similar in composition and curing and have similar as-fabricated tensile modulus and CTE. The data shows that the fabric significantly suppressed crack formation, as compared to the cross-ply laminate and, therefore, exhibited no significant change in the CTE. These data demonstrate the increase in stability of a woven fabric in a thermal cycling environment as compared to the cross-ply laminate fabricated from unidirectional plies.



SYNERGISTIC EFFECTS OF THERMAL CYCLING AND RADIATION ON MICRODAMAGE IN COMPOSITE MATERIALS

Spacecraft materials are not subjected only to thermal cycling. At GEO, the material can also be exposed to significant dose levels of electron radiation. The effects on induced microcracking in two thermoset and thermoplastic Gr/Ep composites of (1) thermal cycling only and (2) irradiation followed by thermal cycling, ref. 3, are shown in the figure below. Quasi-isotropic laminate, [0/45/-45/90]_s, specimens of each material were exposed to either thermal cycling only or 10¹⁰ rads of electron radiation followed by thermal cycling. When these materials were subjected to only thermal cycling, the toughened epoxy laminate T300/BP907 and the thermoplastic laminate AS4/PEEK, had few or no cracks after 500 cycles between -150°C and 93°C. When exposure to electron irradiation was followed by thermal cycling, the T300/BP907 had a very large increase in the microcrack density whereas the AS4/PEEK had a small increase in the microcrack density. The increase in the microcrack density induced by thermal cycling after irradiation is attributed to radiation-induced embrittlement of the matrix.

(500 THERMAL CYCLES BETWEEN 93°C AND -150°C)

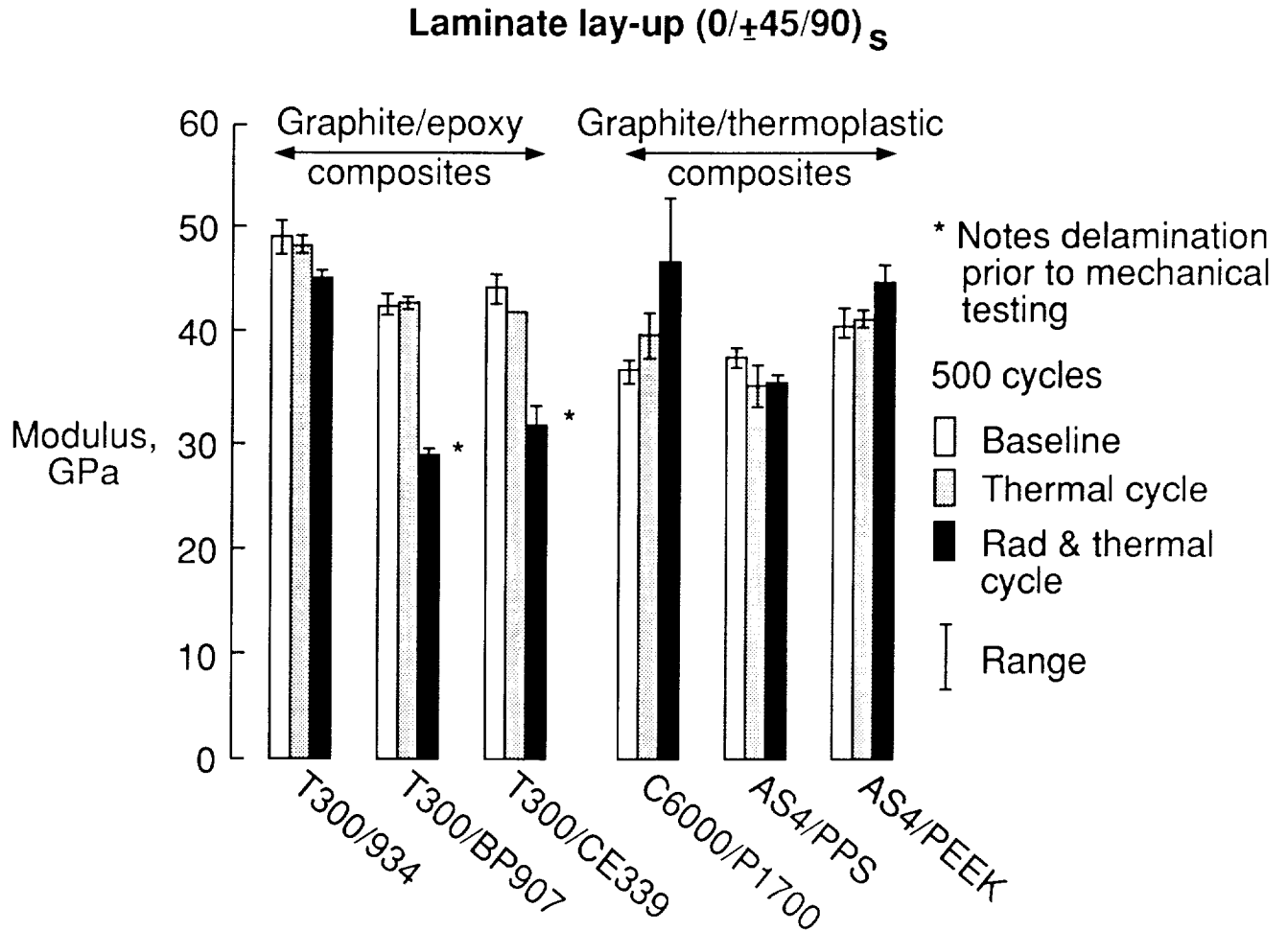
MATERIAL	MICROCRACK DENSITY, CRACKS/CM		
	THERMAL CYCLED	10 ¹⁰ RADS AND THERMAL CYCLED	
T300/934	7	17	
T300 BP907	0	> 50	
C6000 P1700	21	24	
AS4 PEEK	1	5	

→ 1 cm ←

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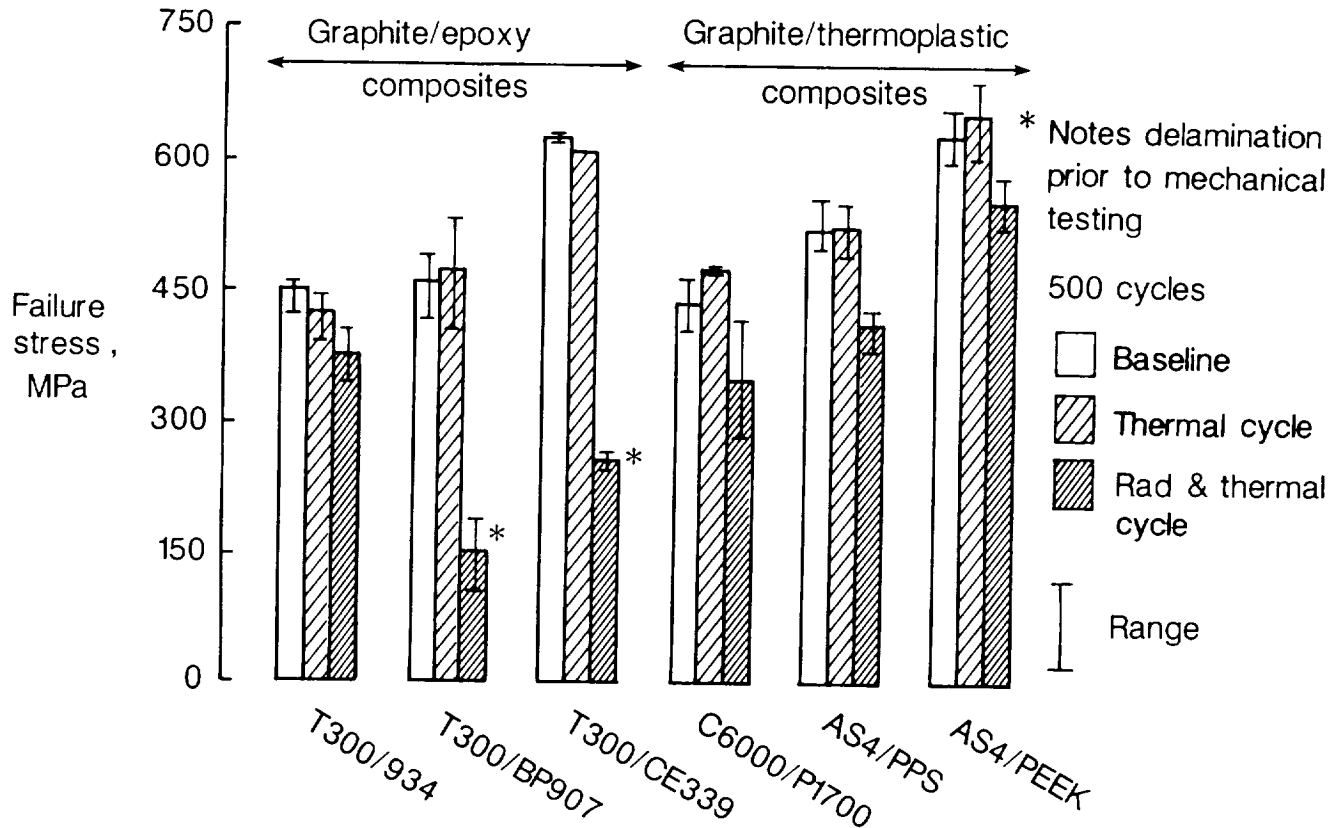
THERMAL CYCLING EFFECTS ON TENSILE MODULUS

The effects of thermal cycling only and radiation exposure followed by thermal cycling on the tensile modulus of three thermoset and three thermoplastic composite materials, ref. 3, are shown below. Thermal cycling only did not significantly change the tensile modulus of a $[0/+45/-45/90]_s$ laminate made with any one of the six materials. Significant reduction in the modulus caused by radiation followed by thermal cycling was observed only in the T300/BP907 and T300/CE339 toughened epoxy laminates. This reduction in both materials is attributed to matrix embrittlement and resulting damage.



THERMAL CYCLING EFFECTS ON TENSILE STRENGTH

The effects of thermal cycling only and radiation exposure followed by thermal cycling on the tensile strength of three thermoset and thermoplastic composite materials, ref. 3, are shown below. (These are the same materials presented in the previous figure.) Thermal cycling only did not significantly change the tensile strength of the $[0/+45/-45/90]_s$ laminate made with each of the six materials. However, radiation exposure followed by thermal cycling resulted in significant reduction in the tensile strength of the three thermoplastic materials and the two toughened thermoset epoxy materials. These data show that there was detrimental synergism between electron radiation and thermal cycling, at least when thermal cycling follows radiation exposure.

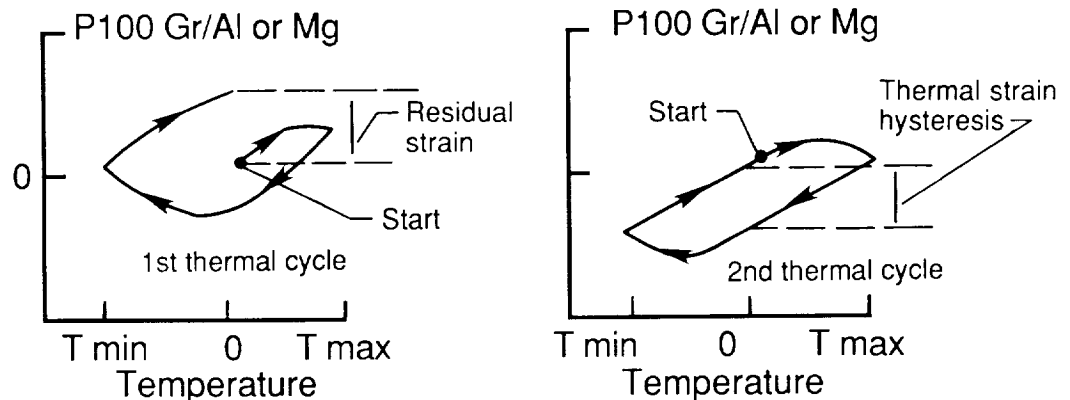


EFFECTS OF THERMAL CYCLING ON METAL-MATRIX COMPOSITES

Advantages of graphite-reinforced metal-matrix composites (MMC) over polymer-matrix composites for space structure applications include higher thermal conductivity and better environmental durability. The standard MMCs considered are continuous graphite-fiber reinforced 6061 Al and AZ91C/AZ61A Mg. However, both of these material systems exhibit thermal strain hysteresis and residual strain during thermal cycling, shown below, which are undesirable for dimensionally critical structures. The response of these materials during thermal cycling is governed by the elastic/plastic deformations of the matrix material, ref. 4. The large residual strain at the end of the first cycle is not seen during subsequent thermal cycles through the same or smaller temperatures ranges because the plastic deformation at the elevated temperatures is offset by the plastic deformation at the low temperatures. There are several potential ways to alter this behavior. The matrix elastic limits could be increased to prevent matrix yielding. Providing the correct matrix alloy chemistry in the final composite could help insure strengthening from heat treatments. Reduction in residual fabrication stresses by thermomechanical treatments could also help reduce total stress levels reached during thermal cycling.

BACKGROUND

- Current state-of-the-art:
 - Standard MMC for space structures: Continuous graphite-fiber reinforced 6061 Al and AZ91C/AZ61A Mg
 - Thermal strain hysteresis and residual strain are recognized problems

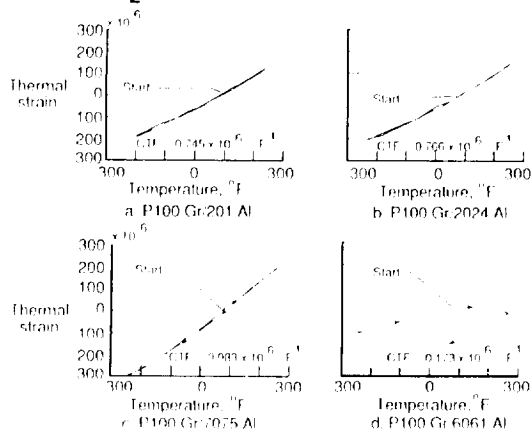


- Potential solutions:
 - Increase matrix tensile and compressive elastic limits
 - Insure correct matrix alloy chemistry
 - Reduce residual fabrication stresses (thermomechanical treatments)

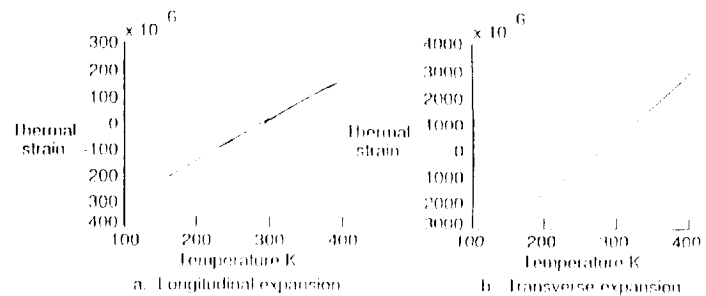
THERMAL EXPANSION BEHAVIOR OF PROCESSED P100Gr/Al [0] COMPOSITES AFTER 1500 CYCLES

The thermal expansion behavior of P100 Gr/Al unidirectional laminates containing high strength aluminum alloys, 201, 2024, and 7075 and the standard 6061, after thermal processing to minimize hysteresis, refs. 4 and 5, is shown in left figure. With the exception of the 6061 Al matrix composite, the P100 Gr/Al laminates show no hysteresis or residual strain even after 1500 cycles. The 6061 Al laminate was not sufficiently strengthened by the heat treatment to eliminate yielding because the chemistry of the alloy was not within specifications. However, when P100 Gr/6061 Al was made with tight control on the chemistry, right figure, processing raised the elastic limit sufficiently to prevent hysteresis. Attempts to heat treat the P100 Gr/Mg MMC with high strength Mg alloys were unsuccessful in eliminating hysteresis. Advancements in high strength Mg alloy development are required to obtain a hysteresis-free P100 Gr/Mg MMC.

THERMAL EXPANSION BEHAVIOR OF THE PROCESSED P100Gr/Al [0] COMPOSITES AFTER 1500 CYCLES



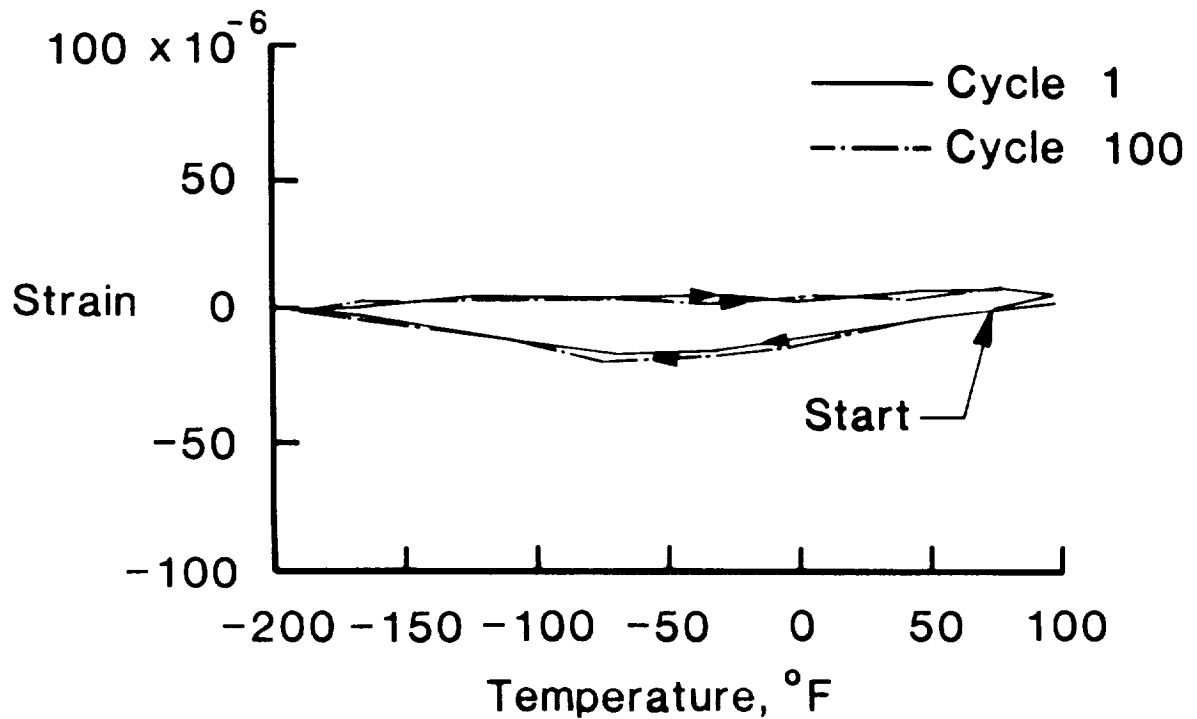
THERMAL EXPANSION OF PROCESSED P100 Gr/6061 Al 1-Ply unidirectional laminate from DOD "Big Buy"



THERMAL EXPANSION OF (0/+ 60)_s HMS/BOROSILICATE GLASS LAMINATE

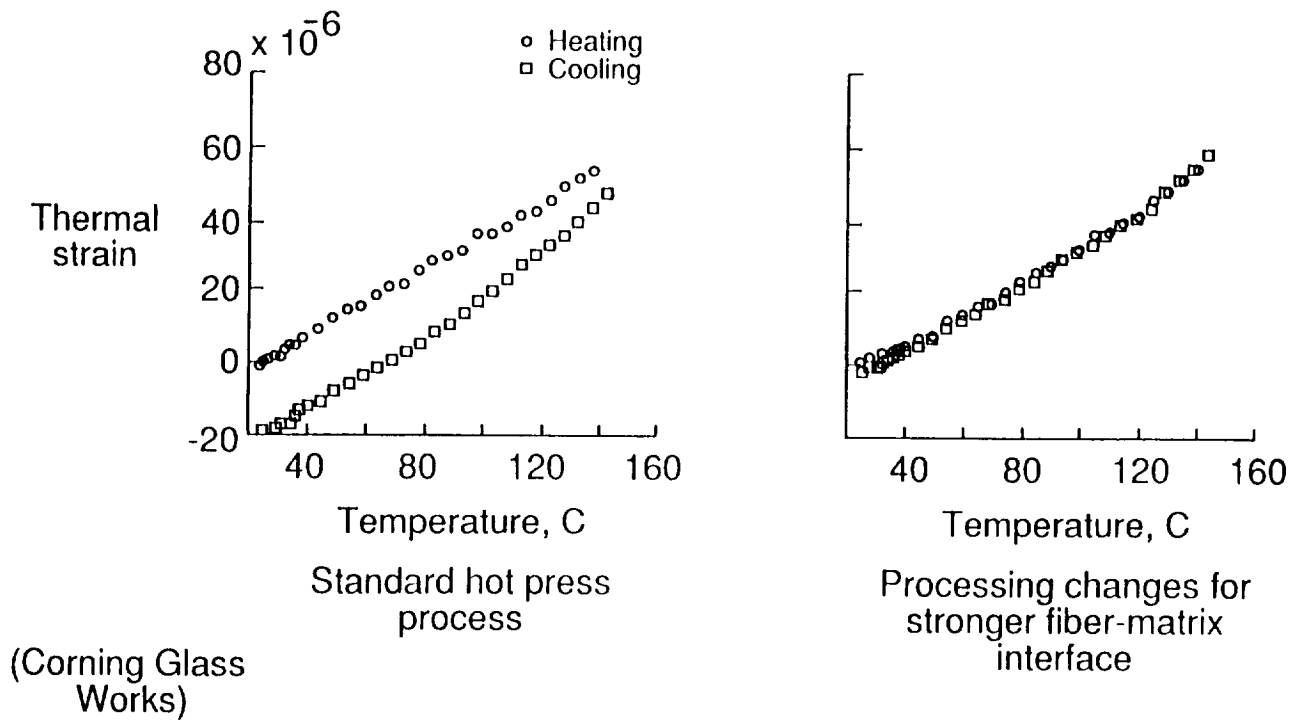
Because of its environmental stability and low outgassing characteristics, graphite-reinforced glass materials are very important for space applications. However, the thermal expansion behavior of this material is characterized by hysteresis which is detrimental to the performance of a precision structure. The thermal expansion of one Gr/glass, HMS/Borosilicate glass, ref. 6, is shown below. The average CTE for this quasi-isotropic laminate is about zero. The expansion behavior before and after 100 cycles exhibits a small hysteresis which is not affected by thermal cycling. The exact mechanism causing this hysteresis is not known but may be the result of the fibers slipping in the glass matrix.

X-Direction



THERMAL EXPANSION OF Gr/ABS GLASS COMPOSITE [0/90]_s LAMINATE

The thermal expansion of a Gr/ABS glass cross-ply laminate before and after processing, ref. 7, is shown below. When the standard hot press process is used, the material exhibits hysteresis when heated from room temperature to about 150°C and cooled back to room temperature. However, when the material is processed in a way to strengthen the fiber-matrix interface, the hysteresis is eliminated. Additional work is needed to investigate the response of the material to verify this behavior and also to determine the behavior when cycled at temperatures below 70°F.



Thermal strain hysteresis eliminated by changes in processing

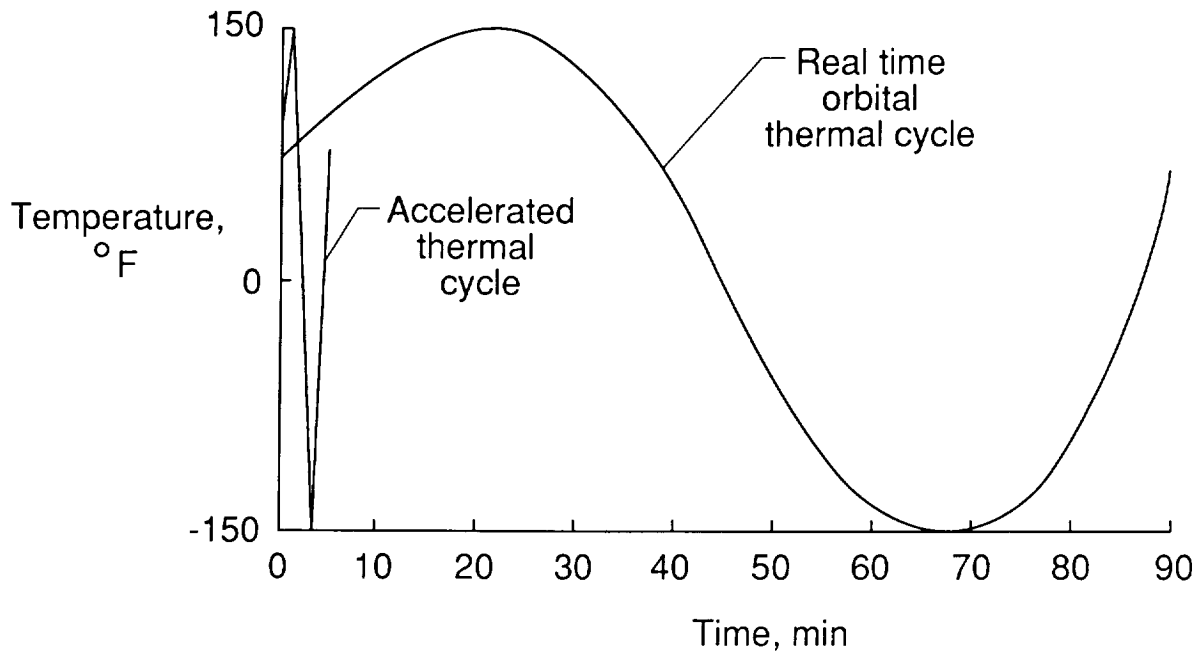
MISSION LIFE VS THERMAL CYCLES

The number of thermal cycles a spacecraft will experience can vary from a little less than 6000 for a one-year life expectancy to as many as 175,000 for a 30-year Space Station. These large numbers of thermal cycles make real time testing impractical. Accelerated testing techniques are required to verify material life assessment and prediction methodologies.

Mission life	No. of 90-min cycles
1 yr.	5840
5 yrs.	29200
10 yrs.	58400
20 yrs.	116800
30 yrs.	175200

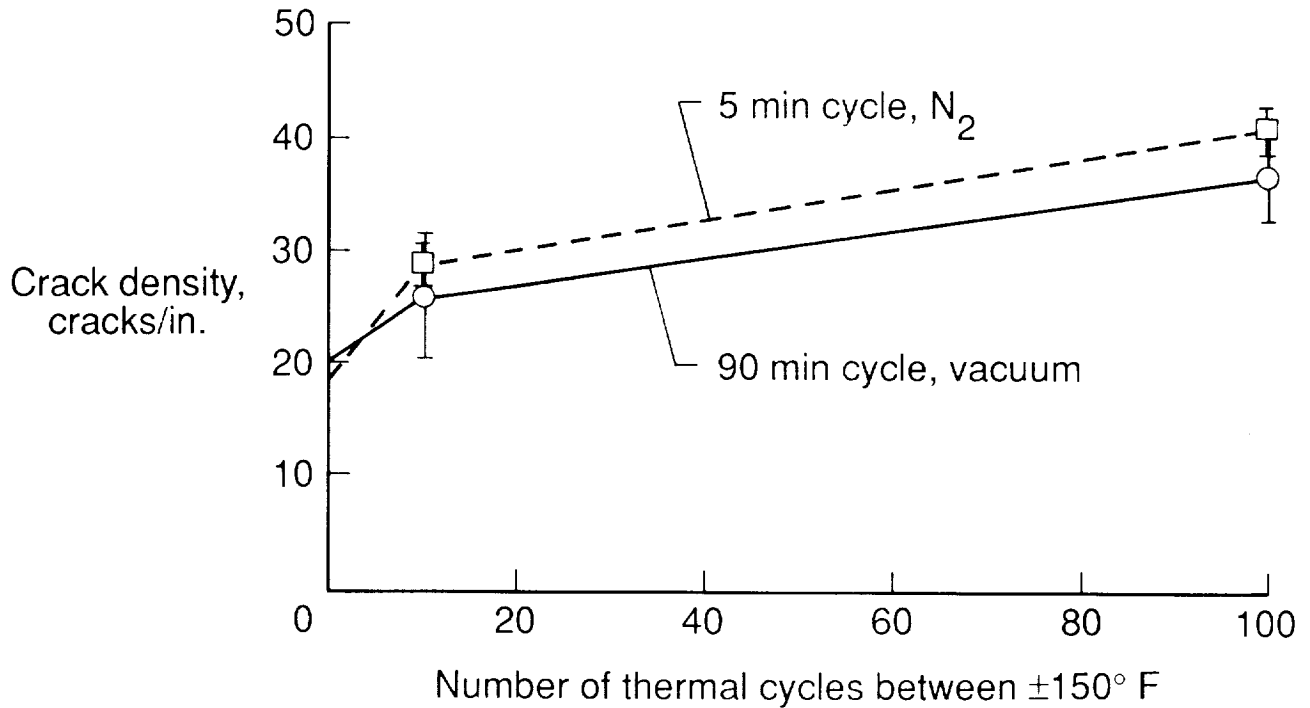
COMPARISON OF ACCELERATED AND REAL TIME THERMAL CYCLES IN SPACE

A comparison of a real time thermal cycle for low Earth orbit and a proposed accelerated thermal cycle is shown in the figure below. The real time cycle takes place in vacuum and has a 90-minute period with an amplitude of $\pm 150^{\circ}\text{F}$. The proposed accelerated cycle has a period of about 5 minutes and an amplitude of $\pm 150^{\circ}\text{F}$. In the accelerated test, a dry nitrogen atmosphere was used instead of vacuum. Heating was therefore done by convection instead of by pure radiation. Composite material samples were exposed to these two environments and the effects on the different environments on a P75 Gr/934 Ep laminate were compared.



EFFECTS OF THERMAL CYCLE RATE AND ATMOSPHERE ON MICROCRACKING IN [0/90/0/90]_s P75 Gr/934 Ep LAMINATE

The validity of the proposed accelerated test technique was established by comparing the microdamage induced in a P75 Gr/934 Ep laminate by a 90-min. cycle in a vacuum and by a 5-min. cycle in nitrogen between $\pm 150^{\circ}\text{F}$, ref. 8. The [0/90/0/90]_s laminate was selected because previous tests have shown that damage was induced in this laminate when it was thermally cycled between $\pm 150^{\circ}\text{F}$. Crack densities in specimens cycled in the slow and accelerated thermal cycles are shown below. The data show that there were no significant differences between the damage induced in this laminate by the two different thermal cycle environments over 100 cycles. This is consistent with similar data obtained by the European Space Agency, ref. 9.



SUMMARY

A brief description of the effects of the space thermal cycling environment on structural composite materials has been given and is summarized below. The material surface optical thermal parameters were shown to have significant effects on the temperature range. The interaction of the environment and the material and the effects of this interaction on resin-matrix (RMC), metal-matrix (MMC), and glass-matrix (GMC) composites were shown and discussed. The primary problem associated with thermal cycling is the microdamage caused by induced thermal stresses. Data were presented that showed accelerated ground tests can simulate the effects of thermal cycling expected in space. The amplitude of the thermal cycle, and therefore the stress levels, can be significantly varied by using a thermal control coating. The effects of thermal cycling are different for the different classes of composite materials. Cycling-induced cracks in RMC caused property changes, and synergistic effects with electron radiation were significant. The MMC exhibited dimensional instability because of plastic deformation. The GMC also exhibited microcracking and thermal strain hysteresis. However, potential solutions to these undesirable effects have been developed for all of the composite material systems studied.

- Primary problem: microdamage caused by induced thermal stresses
- Accelerated ground tests simulate effects on composite materials
- Thermal control coatings will alter the amplitude of the thermal cycle
- Different effects on different classes of materials:
 - RMC - cracking, property changes, synergistic effects with radiation
 - MMC - thermal strain hysteresis, plastic deformation
 - CMC - thermal strain hysteresis, microcracking
- Demonstrated solutions have been developed

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